N 70 34312

NASA TECHNICAL TRANSLATION NASA TT F-12,982

NASA TT F-12,982

INDIRECT METHOD OF DETERMINING AIR FLOW THROUGH INSIDE CHANNEL AND GAS TEMPERATURE IN FRONT OF DOUBLE FLOW TURBINE

N. V. Pervyshin, V. N. Denisov

Translation of "Kosvennyy Metod Opredeleniya Raskhoda Vozdukha Cherez Vnutrenniy Kontur i Temperatury Gaza Pered Turbinoy Dvukhkonturnogo Dvigatelya" Izvestiya Vysshikh Uchebnykh Zavedeniy, Seriya "Aviatsionnaya Tekhnika," No. 1, 1969, pp. 93-98

CASE FILE COPY

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546 MAY 1970

SYMBOLS

T^*_g -- temperature of decelerated flow at turbine intake, °K;

P^*_g -- pressure of decelerated flow ahead of turbine, n/m^2;

G_g -- gas flow through high pressure turbine, n/sec;

G_{aI} -- air flow through inner channel of double flow engine, n/sec;

K_l -- coefficient that takes into account the consumption of air for cooling the vanes of the turbine and losses through the labyrinths;

G_f -- the per second fuel consumption in the combustion chamber, n/sec;

i^*_g -- the heat content of the gas in the decelerated flow in front of the turbine, j/kg;

i^*_{hpc} -- heat content of the gas in the decelerated flow behind the high-pressure compressor;

H_u -- lower calorific capacity of the fuel, j/kg;

η_z -- the coefficient of completeness of combustion in combustion chamber;

G_{ahp} -- the flow of air through the high-pressure compressor, corrected to standard atmospheric conditions with respect to the parameters at the high-pressure compressor intake, n/sec;

n_{rhp} -- the number of revolutions of the high-pressure rotor, corrected for the conditions at the intake of this compressor, rpm;

n_{lp} -- number of revolutions of the low pressure rotor in rpm;

ε^*_{hpc} -- degree of increase of pressure in high-pressure compressor;

σ_{cc} -- coefficient of recovery of total pressure in combustion chamber;

α -- coefficient of excess air;

$P_{ln.a}$ -- the area of the first nozzle apparatus in the cold state, m^2.

INDIRECT METHOD OF DETERMINING AIR FLOW THROUGH INSIDE CHANNEL AND GAS TEMPERATURE IN FRONT OF DOUBLE FLOW TURBINE

N. V. Pervyshin, V. N. Denisov

ABSTRACT. An indirect method is presented for determining the air flow through the inner channel of a double flow turbojet engine and for measuring the gas temperature in front of the turbine on the basis of measurements of temperature and pressure behind the high-pressure compressor and fuel flow through the engine; this method possesses universality. The method can also be used for determining the air flow and temperature for turbojet (single flow) engines in which the first stage of the turbine operates at the critical pressure drop.

The flow of air in modern turbojet engines is determined on the basis of /93* measurements of total and static pressures, temperature in the intake channel of the engine, and subsequent calculation of the velocity fields. In view of the difficulty of the tests and processing of the experimental results, a special measuring intake apparatus is sometimes used (it may also be the intake channel of the engine), by means of which the air flow can be found on the basis of pressure drops measured at one or several points around the channel, and on the basis of calibration curves. Attempts to determine the flow of air on the basis of temperature and pressure measurements behind the compressor or turbine have been unsuccessful, since the flow in these cross sections is much more uneven than in the intake channel, thus precluding the measurements of sufficient precision.

The temperature of the gases in front of the turbine can be measured by thermocouple manifolds installed in front of the turbine or located in the vanes of the nozzle apparatus. However, this type of measurement can be made /94 on individual, specially designed engines.

In the majority of cases the gas temperature in front of the turbine is found from the equation of thermal balance, the solution of which requires the

*Numbers in the margin indicate pagination in the foreign text.

knowledge of the air and fuel flows, air temperature behind the compressor, lower calorific capacity of the fuel, and coefficient of combustion completeness.

Thus the determination of the air flow and gas temperature in front of the turbine of turbojet engines (TJE) is not especially difficult. As regards double flow engines, however, it is a different matter.

By using the proposed method it is possible to find the total air flow through a double flow engine, but the air flows through the first and second stages remain unknown. Without knowing the air flow through the inner channel of the engine it is impossible to use the thermal balance equations. It is also impossible to take measurements in any cross section of the second channel for the purpose of determining the air flow through the second channel for the very same reasons as for measurements in cross sections behind the compressor or turbine of a TJE.

To determine the desired values, indirect methods that provide a sufficient degree of precision can be used, using for this purpose, instead of the thermal balance equation, other equations that take into account the features of operation of the inner channel of a double flow engine. All modern double flow engines with forward ventilators (see Figure 1) are double cascade engines. It can be shown that the position of the point of joint operation on the characteristic of a high-pressure turbine (h.p.) of such an engine theoretically does not depend on the mode of operation. Consequently, the throughput capacity of the high-pressure turbine, characterized by the parameter $G_g\sqrt{T^*_g}/P^*_g$ (actually this parameter will depend on the gas temperature in front of the turbine due to the thermal expansion of the turbine parts and the associated change in the passage areas) also does not depend on the mode of operation.

Since the gas pressure in front of the turbine and the air pressure behind the compressor are related by the equation

$$P^*_g = \sigma_{cc} \cdot P^*_k,$$

and the gas flow through the turbine is

$$G_g = G_{aI}(1 - k_1) + G_f,$$

where σ_{cc} is the coefficient of pressure recovery in the combustion chamber; /95

k_1 is the coefficient that takes into account the consumption of air for cooling the turbine blades and losses in the labyrinths;

G_f is the per second fuel consumption in the combustion chamber, we may then write:

$$\frac{[G_a(1-k_1)+G_f]\cdot\sqrt{T^*_g}}{\sigma_{cc}P^*_c} = A = \text{const.} \qquad (1)$$

The values k_1 and σ_{cc} in equation (1) are, for all practical purposes, independent of the mode of operation of the engine. In the tests the pressure behind the compressor and the fuel flow are measured while the values k_1 and σ_{cc} are known from the results of special tests of the respective systems. Then only the air flow and temperature of the gases in front of the turbine remain unknown in formula (1). For that reason determination of these values requires another equation, for which we may use the thermal balance equation:

$$i^*_g = i^*_{hpc} + \frac{G_f}{G_g}\cdot H_u\cdot\eta_z. \qquad (2)$$

Here i^*_g is the heat capacity of the decelerated flow in front of the turbine;

i^*_{hpc} is the heat capacity of the decelerated flow behind the high-pressure compressor, determined on the basis of the air temperature behind the compressor, measured during the tests;

H_u is the lower calorific capacity of fuel;

η_z is a coefficient that takes into account the combustion completeness in the combustion chamber, which is determined in leakage tests or in tests of the entire chamber on special test stands.

The joint solution of equations (1) and (2) enable us to find the temperature T^*_g and air flow through the inner channel G_a.

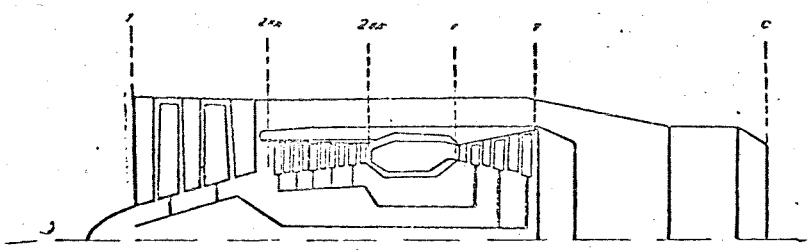

Figure 1. Diagram of Double Flow Turbojet Engine.

The dependence of the throughput capacity A on area $F_{ln.a}$ can be established either by calculations, or experimentally by blowing (through-running) the first nozzle apparatus. Tests were made of an insulated high-pressure cascade, which represents a single-cascade TJE, while one of the Soviet double flow engines was being revved up. In this case the throughput capacity A was determined by the direct change of the air flow through the high-pressure cascade and by calculating the temperature T_g^* by ordinary methods. A series of such tests of an insulated high-pressure cascade with first nozzles of different areas made it possible to establish the dependence of A on $F_{ln.a}$, which was then used for calculating T_g^* and G_a during testing of a full scale double flow engine. The area of the nozzle apparatus of the turbine is measured in the cold state. As the temperature of the gases in front of the turbine changes, the linear dimensions of the rotating and fixed parts of the turbine will also change, and consequently, so will the area of the cross sections and A.

On the basis of the processing of a large quantity of test results from the high-pressure cascade, the dependence of the ratio A/A_{max} on the relative gas temperature in front of the turbine $T_g^*/T_{g\ max}^*$ was determined for the double flow engine and is represented in Figure 2.

Since the solution of equations (1) and (2) requires the use of the series approximation method, the nomogram shown in Figure 3 is constructed to simplify the calculations. This nomogram can be used for any gas turbine engine with a critical pressure drop in the first stage of the turbine and is constructed for the constant combustion completeness coefficient, equal to 0.98. For other values of η_z, similar nomograms can be constructed. For most engines the

/96

coefficient of combustion completeness can be assumed constant in a wide range of conditions. Figure 4 shows the change in the relative combustion completeness coefficient as a function of the relative excess air coefficient for one of the double flow engines.

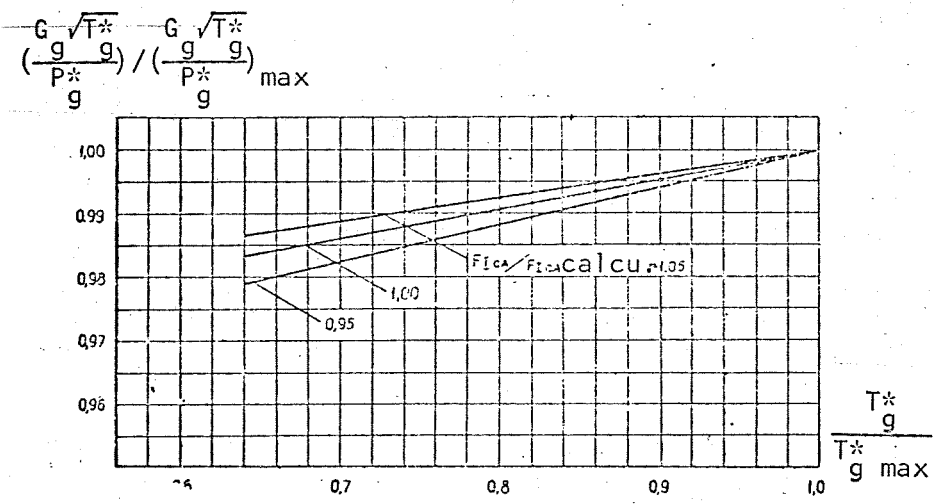

Figure 2. Relative Throughput Capacity of High-Pressure Turbine as Function of Area of First Nozzle Apparatus and Their Temperature Ratio $T^*_g/T^*_{g\,max}$.

This function can change under flight conditions, in particular, in engine operating modes close to the rare gas mode. In this case the calculations are made for other values of η_z.

The range of operating modes under surface static conditions are plotted on the curve of the dependence. The coefficient of combustion completeness changes by about 0.5% in the entire operating range. Calculation of the gas temperature T^*_g with the aid of the nomograms is accomplished in the following manner:

the set of parameters

$$G_g/A \cdot P^*_g$$

is calculated;

knowing the decelerated value of the temperature behind the high-pressure compressor T^*_c and the set $G_f/A \cdot P^*_g$, the temperature T^*_g is calculated with the

aid of the nomogram;

the air flow through the inner channel G_{aI} is calculated by equation (1), by substituting the value of T^*_g found.

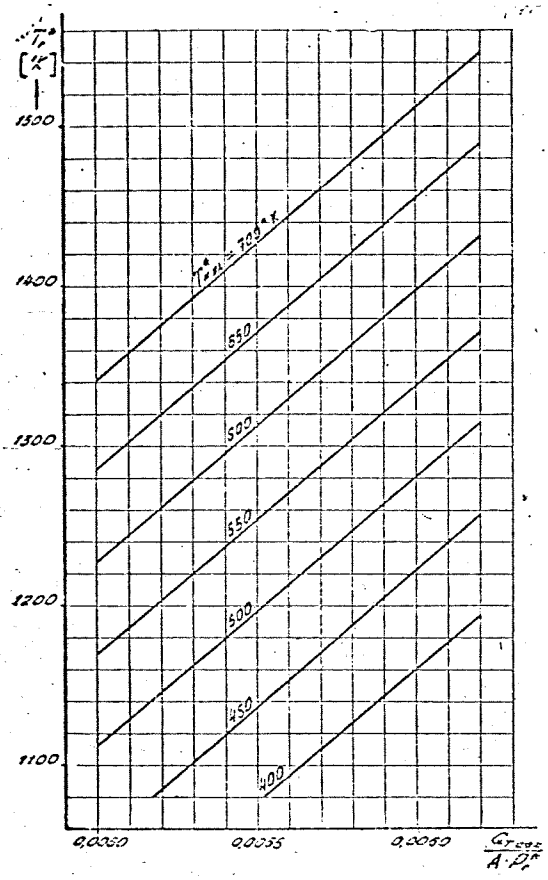

Figure 3. Sample Nomogram for Determination of Gas Temperature in Front of Turbine.

The method described is tested on a double flow engine[1].

For the purpose of measuring T^*_g /98 on 30 vanes of the first nozzle apparatus, distributed equally around the channel, 5 thermocouples were placed on each vane through the height of the channel (a total of 150 thermocouples).

Comparison of the results obtained is shown in Figure 5, from which we see that they coincide satisfactorily. In the rpm range of the rotor of the low-pressure cascade from maximum to 0.77 maximum, the measured gas temperatures in front of the turbine and the temperature of the gases in front of the turbine as calculated by the above-described method, practically coincided. Only for a low-pressure cascade rotor rpm of less than 0.6 as a maximum did the difference between the measured and calculated gas temperatures in front of the turbine amount to as much as 2%.

The indirect method presented herein for determining the air flow through the inner channel of a double flow turbojet engine and for measuring the gas temperature in front of the turbine on the basis of the measurements of

[1]Engineers I. P. Anisimov, V. S. Kondrusev and V. A. Lipov participated in the tests.

temperature and pressure behind the high-pressure compressor and fuel flow through the engine possesses universality. It can be used under operating conditions aboard an aircraft, in an airborne laboratory, in a thermobaro chamber, and even for special tests of engines, for instance, with an aircraft intake apparatus, with throttling grids, and with an air heater in the engine intake. Furthermore, the indirect method proposed herein can be used for determining the air flow and temperature T_g^* for turbojet (single flow) engines in which the first stage of the turbine operates at the critical pressure drop.

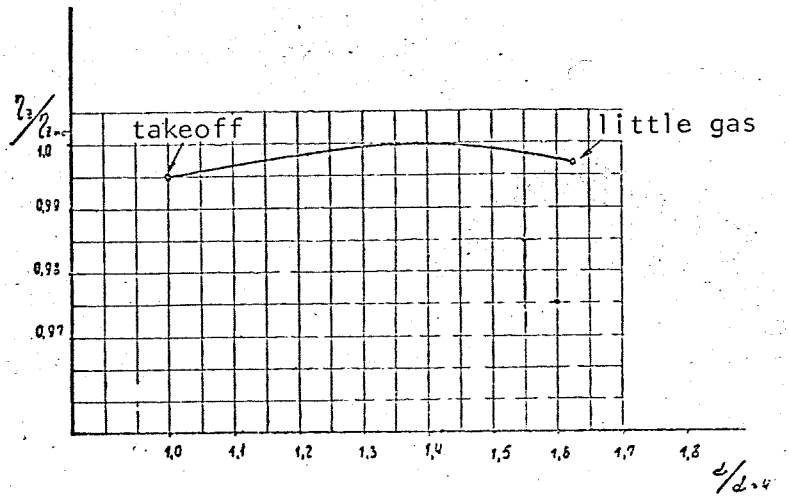

Figure 4. Relative Combustion Completeness Coefficient as Function of Relative Excess Air Coefficient.

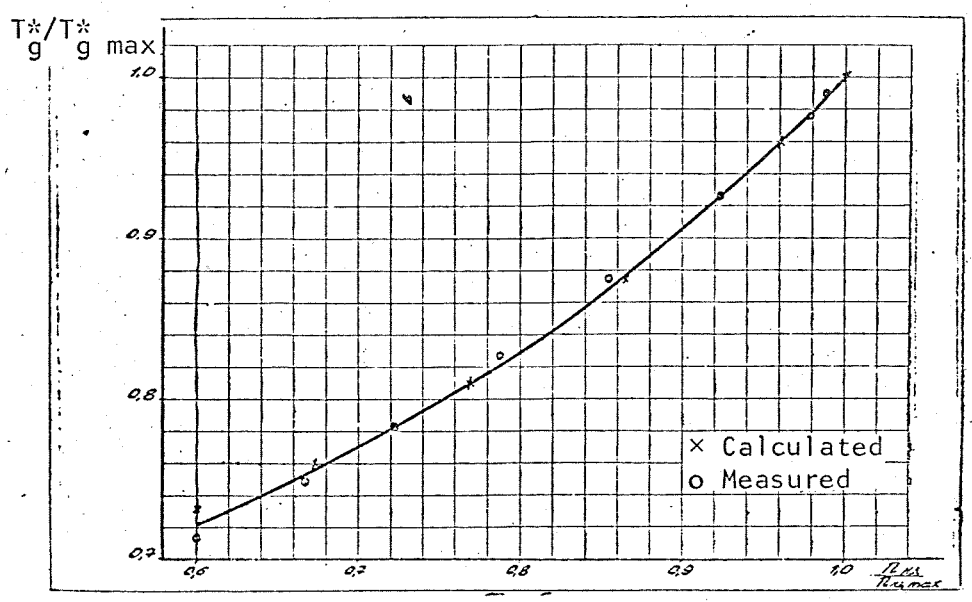

Figure 5. Comparison of Measured and Calculated Values of Gas Temperature in Front of Turbine.

Translated for the National Aeronautics and Space Administration under Contract No. NASw-2037 by Techtran Corporation, P.O. Box 729, Glen Burnie, Md. 21061